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A SONIC-FLOW ORIFICE PROBE FOR THE IN-FLIGHT MEASUREMENT OF TEMPERATURE PROFILES OF A JET ENGINE EXHAUST WITH AFTERBURNING

By C. Dewey Havill and L. Stewart Rolls

Ames Aeronautical Laboratory
Moffett Field, Calif.

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SUMMARY

A temperature-measuring system using two sonic-flow orifices in series is described. This system was adapted to measure the exhaust gas temperature of an afterburning jet engine and was mounted in a swinging pitot-static probe. The sonic-flow orifice system had only a small variation in temperature at the thermocouple and temperature change was measured as a change in pressure. The characteristic time lag of the sonic orifice system was 0.045 second. Measurement of exhaust temperatures of 3200° R was obtained during afterburner operation in flight.

INTRODUCTION

During the flight tests of prototype aircraft an accurate determination of the airframe drag is frequently desired. To measure the drag in flight it is necessary to determine the actual propulsive force existing on the airplane. This thrust determination is complicated with high-power turbojet engines using afterburners for thrust augmentation. The necessity of providing cooling air for both airframe and engine has resulted in complex engine installations. It is very difficult either to assess the actual performance of the engine or to investigate performance losses caused by the cooling air flow, without knowledge of the total exit mass flow and momentum.

The total jet-exit mass flow may be obtained by two means. The first method would be to instrument all inlets and sum up all these individual mass flows, while the second method would be to determine the mass flow at the fuselage exit plane by measuring the pressure and temperature distribution. The first method requires a larger amount of instrumentation and errors could result from leakage between the measuring station and the exit. As a result of these difficulties, a complete fuselage exit survey is used at the Ames Aeronautical Laboratory. This method is based on temperature and pressure survey at the fuselage exit (ref. 1) obtained by use of a swinging probe traversing the jet region. (See fig. 1.)

The method outlined in reference 1 is a reliable means for determining the thrust of an installed engine in flight; however, the measurement of the rapid temperature rise at the edge of the jet with the swinging probe required a thermocouple with zero time lag capable of withstanding afterburner temperatures. To use the shielded chromel-alumel thermocouple system described in reference 1, it was necessary to use ground-determined response characteristics in the workup of the data and to apply large corrections to the measured temperature to obtain the true temperature. Since there was some doubt about the precision of these corrections, a device which allowed a more direct measurement of the temperature was developed. This device is similar to that described in reference 2. Two sonic-flow orifices operate in series such that the stream temperature change is measured as a change in pressure. The necessary reference temperature measurements are made in the region of relatively constant temperature.

The purpose of this report is to discuss the use of the sonic orifice probe to determine the temperature variation across a jet engine exhaust with afterburning and to present some results illustrating use of this probe.

NOTATION

A	orifice area, sq ft
D	diameter of fuselage exit, ft
h_p	flight pressure altitude, ft
k	constant
M	flight Mach number
P	total pressure in tail pipe, lb/sq ft
P_T	total pressure at orifice, lb/sq ft
R	gas constant, 1715 sq ft/^{o}sec^{2}
r	distance from jet center line to probe location, ft
T	total temperature, ^{o}R
V	volume in the sonic flow chamber, cu ft
w	mass flow through the orifice, slugs/sec

ρ density of gas, slugs/cu ft

γ ratio of specific heats

Subscripts

1 orifice 1

2 orifice 2

i initial conditions

fs free stream

tp tail pipe

DESCRIPTION

The principle upon which the operation of the sonic-flow orifice probe is based is presented in reference 2. This analysis states that for sonic flow through an orifice:

$$\frac{P_T A}{w\sqrt{RT}} = \frac{1}{\sqrt{\gamma}}\left(\frac{\gamma + 1}{2}\right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \tag{1}$$

Evidence presented in reference 2 indicated that for a given gas the value of γ may be assumed constant. If the gas constant (R) is also assumed to be constant, then equation (1) may be written:

$$w = k\frac{P_T A}{\sqrt{T}} \tag{2}$$

where k is a constant. Thus, it will be noted that for a given orifice of area A, if the mass flow, w, and total pressure, P_T, are known, it is possible to calculate the total temperature, T. When two such orifices are placed in series so the same mass flow must pass through both orifices, it is possible to equate the equation (2) for each orifice and combine them to obtain the following:

$$k_1\frac{P_{T_1} A_1}{\sqrt{T_1}} = k_2\frac{P_{T_2} A_2}{\sqrt{T_2}}$$

or

$$T_1 = T_2 \left(\frac{P_{T_1}}{P_{T_2}}\right)^2 \left(\frac{k_1 A_1}{k_2 A_2}\right)^2 \tag{3}$$

It will be noted in equation (3) that an unknown gas temperature can be determined from conditions measured at the second orifice, provided the effective area ratio, $\left(\frac{k_1 A_1}{k_2 A_2}\right)$, can be calibrated. It is worthwhile to note such a device can be made relatively free from radiation and conduction errors, normally associated with temperature measurements, by maintaining the temperature T_2 at a low value (in these tests about 750° R).

A drawing of the sonic-flow chamber, with its two sonic orifices, developed to measure the jet temperature is shown in figure 2. Photographs of the probe showing the type of construction are shown in figure 3. The probe as shown in these figures also includes the total- and static-pressure orifices for the standard survey probe; thus, the three measurements are essentially at the same radius of the jet. The total pressure orifice for the jet and the first orifice of the sonic chamber are located on the spherical nose of the probe in the same vertical plane and equally spaced 15° above and below the stagnation point. The following requirements were incorporated in the probe design to assure the conditions of equation (3) were met: (a) a small volume was maintained between the two orifices to insure rapid response during large changes of pressure and temperature ahead of the first orifice, (b) orifice sizes were selected to assure sonic flow through both orifices with the pressure ratios available, (c) cooling of the gas between the two orifices was provided so the temperature variation at the second orifice would be small and could be measured with a standard thermocouple, and (d) a pressure transducer of small internal volume was used to measure the total pressure at the second orifice as the transducer volume acts as a source or a sink and hence the mass flow through both orifices is not the same during rapid pressure changes.

The response time of the thermocouple is relatively slow so the amount of lag in the system can be reduced by minimizing the temperature change to which it is subjected. In an endeavor to keep the temperature change measured by the thermocouple at the second orifice as small as possible, a heat-source-type cooler consisting of a honeycomb metal block was mounted within the chamber. In this installation, the maximum change in the thermocouple temperature was less than 10° R. Since the operation of the sonic-flow device is based on mass flow and not volume flow, the gain or loss of heat between the two orifices does not invalidate the equations used to compute the jet temperature.

The speed of response of the sonic-flow thermometer is dependent on V the volume between the two orifices (including the volume associated with the pressure-measuring device). The characteristic time lag of the instrument, based on a discussion in reference 3, may be expressed as:

$$\frac{\rho_1 V T_1}{k P_1 \sqrt{T_2}}$$

Solving the above expression for the conditions used in this investigation indicates a characteristic time lag of 0.0063 second during afterburner-off operation. Combining this lag with the over-all lag of the pressure-measuring system would indicate a maximum lag of 0.045 second for the sonic orifice probe.

INSTRUMENTATION AND ASSOCIATED EQUIPMENT

To measure the total pressure at the second orifice, a modified Statham flush pressure transducer (model P-81) was mounted at the top of the swinging arm and connected to the sonic-flow chamber by a 1/32-inch hypodermic tubing. The modification to the transducer consisted of mounting a chamber of near zero volume (the internal height of the chamber was 0.003 inch) above the flush side of the transducer which was then connected to the hypodermic tubing.

The temperature of the gas at the second orifice was measured by an iron-constantan thermocouple mounted just behind the orifice. The thermocouple was mounted at this position to avoid the difficulties of sealing the chamber if the thermocouple wires were brought through the chamber walls. The output from the thermocouple was connected to a miniature self-balancing potentiometer (ref. 4) and the resulting signal recorded on an oscillograph.

Standard NACA photographic recording instruments supplemented with a Consolidated 36-channel oscillograph and correlated by a common timer circuit were used to record the required measurements during a test run. The recording system for the total and static pressures was the same as that described in reference 1. A 12-1/2-foot nose boom was mounted on the test airplane to obtain reliable values of free-stream quantities.

To assure a sonic velocity through both orifices, the tube leading from the sonic-flow chamber was connected to a standard laboratory type high-vacuum pump. In the present installation in an F-94C airplane, the pump is located in the nose of the airplane and is connected to the probe by a 3/4-inch-diameter tube. An electrically operated valve, placed in the line at a point near the probe, remained closed whenever the probe was inoperative, thus enabling the tube from the valve to the pump to act as a vacuum chamber. This chamber had sufficient volume so that the pressure changes at the second orifice during a probe traverse were small. This pressure was maintained at 70 pounds per square foot absolute. The valve was opened 5 to 7 seconds prior to the start of the probe traverse to assure the establishment of uniform flow through the sonic orifices before the data run was started. The capacity of the pump was sufficient to maintain a pressure ratio across each orifice

of sufficient magnitude to insure sonic flow and be able to overcome any pressure losses in the pressure line running from the pump to the probe.

CALIBRATION

To obtain the effective orifice area ratio, the $\left(\frac{k_1 A_1}{k_2 A_2}\right)$ term in equation (3), a series of runs was made on the ground where all the quantities in the equation were known except the area ratio squared. These tests indicated that the effective area ratio squared for this probe dimension was 0.061. The difference between this value and the geometric area ratio squared (0.086) could be the result of the boundary layer reducing the area of the orifice or of errors in manufacturing since only general machine practice was used during the construction of the probe and the orifice discharge coefficients.

A static calibration of the sonic-flow probe was made in a variable temperature furnace. The probe was mounted so that just the nose of the probe was in the furnace and the outlet of the sonic chamber was connected to a vacuum pump. Since the conditions were static during this calibration, standard laboratory pressure and temperature measuring equipment was used. Certain conditions during the static calibration differed from those during flight tests; namely, the stream velocity ahead of the probe and the temperature gradients along the probe. As mentioned in reference 2 for a stream velocity near zero, a portion of the air entering the first orifice has been drawn from near the probe surface and may be cooled by this surface. Thus, the temperature of the air sampled may be less than the furnace temperature. Because of the temperature gradients along the probe during the furnace calibration, the thermocouple for measuring the temperature of the gas at the second orifice was mounted at the rear end of the cooler, thus it would measure the temperature of the gas more accurately than at the more rearward location. The results of this furnace calibration are presented in figure 4. The agreement between the measured temperature and the furnace temperature (standard deviation of the scatter equals $\pm 10.5^{\circ}$ F) indicates the effect of the aforementioned quantities to be small and therefore the sonic-flow probe can be used satisfactorily as a static-temperature probe.

TYPICAL TEST DATA

The evaluation of the sonic-flow probe as a dynamic temperature probe was made in flight and is based on a comparison of measured temperature and pressure profiles in an exhaust jet. Some typical profiles of total pressure and stagnation temperature are presented in

figures 5 through 8. Figures 5 and 6 are profiles with the afterburner operating at flight Mach numbers of 0.63 and 0.82 at 25,000 feet altitude. These curves show that the temperature and pressure profiles are similar, which is as expected, and indicate an abrupt rise in temperature and pressure at the edge of the jet. These curves indicate a temperature in the afterburner of about 3200° R and the data points indicate a scatter, in the level portion of the curve, of about ±100° R. As there is no other method available to measure the gas temperature in the jet during afterburner operation, to check the values measured by the swinging probe, the free-stream temperature has been spotted on the curves as a verification of the initial and end temperature. These free-stream temperatures agree with those measured by the sonic-flow probe. The temperatures measured by the sonic-flow probe should be a few degrees higher than the free-stream temperature due to the fact that the sonic-flow chamber is measuring the temperature of the boundary-layer air which is heated slightly as it flows aft along the fuselage.

Figures 7 and 8 are profiles of the temperature and pressure measured with the afterburner off at flight Mach numbers of 0.63 and 0.80. These data are similar in shape to those shown in figures 5 and 6 for the afterburner on. The maximum temperature with the afterburner off, however, was only 2000° R. The afterburner-off data indicate an asymmetry in the jet tail-pipe flow on opposite sides of the jet center line which is not dependent on direction of probe traverse. This asymmetry is believed to be an engine characteristic and not a probe deficiency as other tests have indicated jet engines may have asymmetric exhaust distributions.

It will be noted in these figures that as the probe enters the jet (negative values of $r/(D/2)$) there is an abrupt overshoot in the measured temperature. This is attributed to different lag characteristics for the free-stream total pressure measurement and the pressure measurement inside the sonic chamber, P_2. The difference in lag is due to the small size of the first orifice, and can be seen in figure 9(a), which presents pressure profiles obtained from a survey of the jet in both directions. As the overshoot in the measured temperature occurs over a small range of the fuselage exit, it would be possible in practice to fair the data without using these points. Another possible way of overcoming this lag effect would be to swing the probe both directions and use the best portions of both surveys. Temperature profiles of the jet with the probe swung in both directions are presented in figure 9(b).

The sonic-flow temperature measuring device has been used to determine approximately 100 profiles of the type shown in figures 5 through 9. Of these profiles about 25 were with the afterburner on. Based on the tests to date and the known uncertainties of the various components of the sonic-flow temperature-measuring system, the uncertainty of the complete system is estimated to be from 3 to 5 percent of the temperature value.

During the development of the sonic-flow temperature-measuring device, other possible uses for this type of temperature measuring system were discussed. As this system determines the stagnation temperature without the effect of a temperature recovery factor, it could be used, in conjunction with a simple computer for Mach number correction, as a free-stream static-temperature indicator. Another possible use would give a continuous indication of the tail-pipe temperature during afterburner operation. Such an application would require the probe to be mounted on a cooled strut. This type of an indication would allow the pilot to meter the fuel flow to the afterburner to obtain the maximum range potentiality from his available fuel.

CONCLUDING REMARKS

The development of the sonic-flow temperature-measuring probe as described in this report has resulted in a satisfactory method for measuring the temperature distribution in the exhaust of a jet engine with afterburning. A static calibration of the probe in a controlled furnace indicated a standard deviation of the errors to be $\pm 10.5^{\circ}$ F. The characteristic time lag of the system was evaluated to be 0.045 second. This temperature-measuring system improved the reliability and reduced the computational time required of the previously used system. This system also reduced the uncertainty in the net thrust measurements to less than 5 percent.

Ames Aeronautical Laboratory
National Advisory Committee for Aeronautics
Moffett Field, Calif., Mar. 9, 1956

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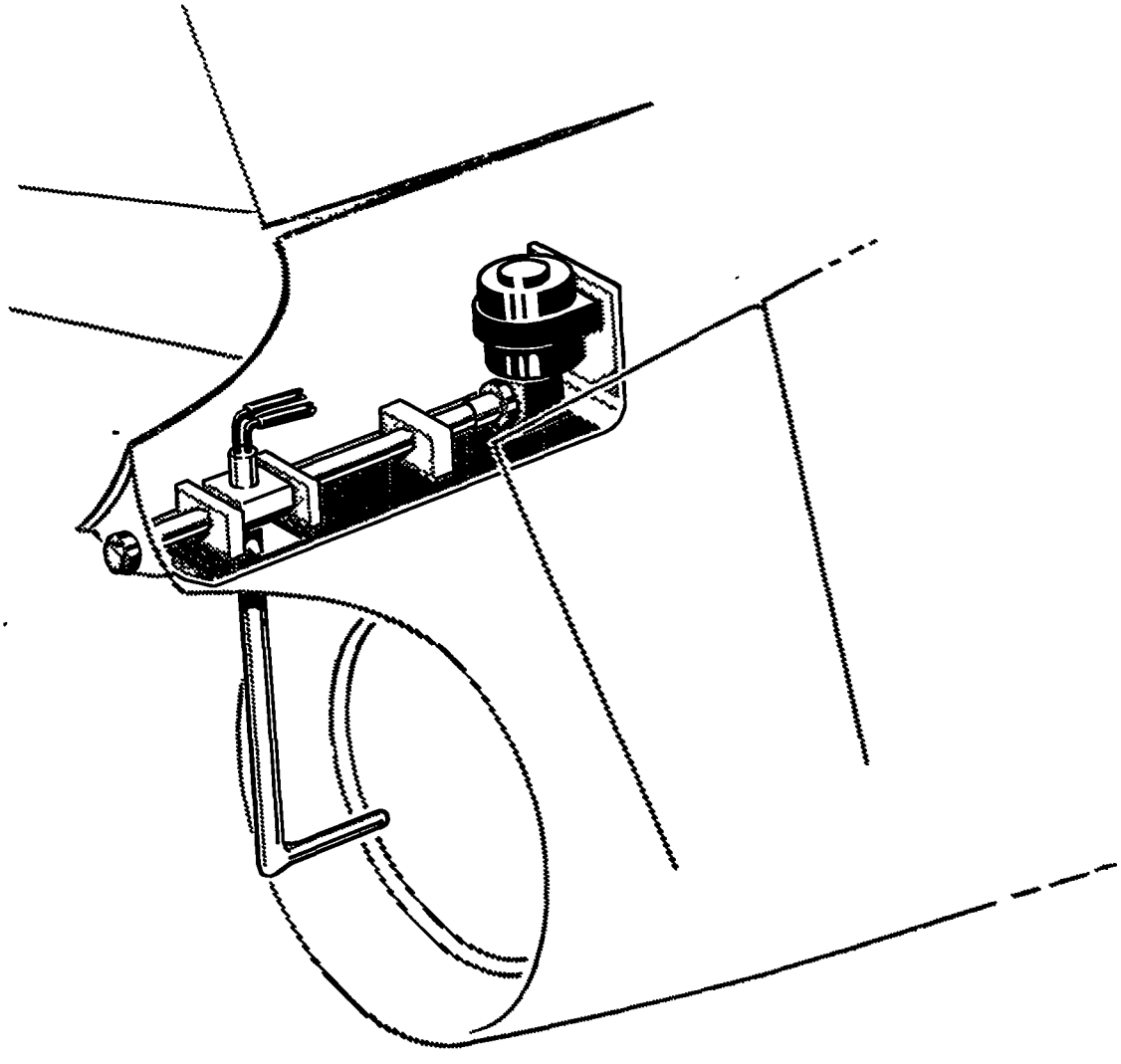


Figure 1.- General view of the Ames swinging-probe installation.

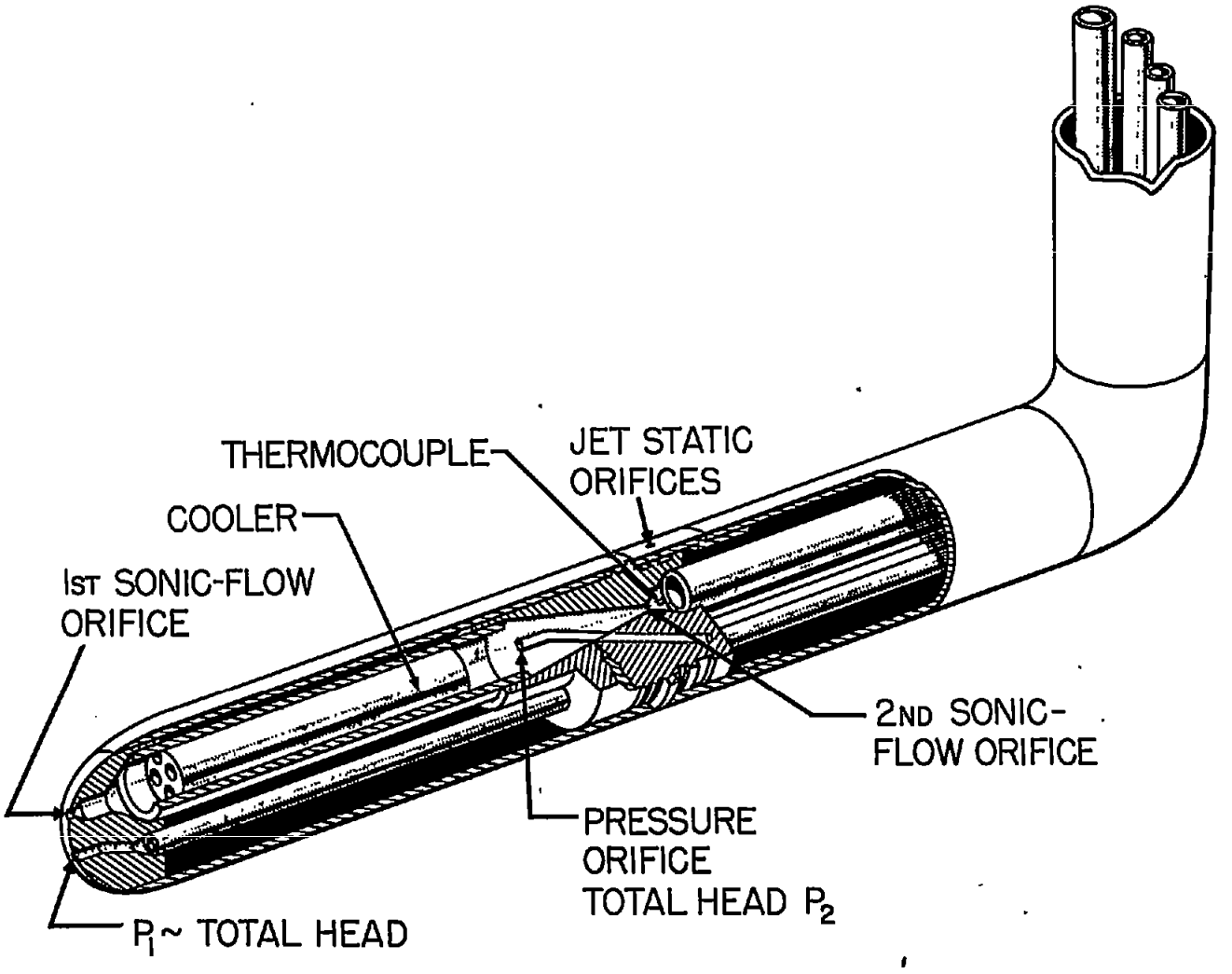
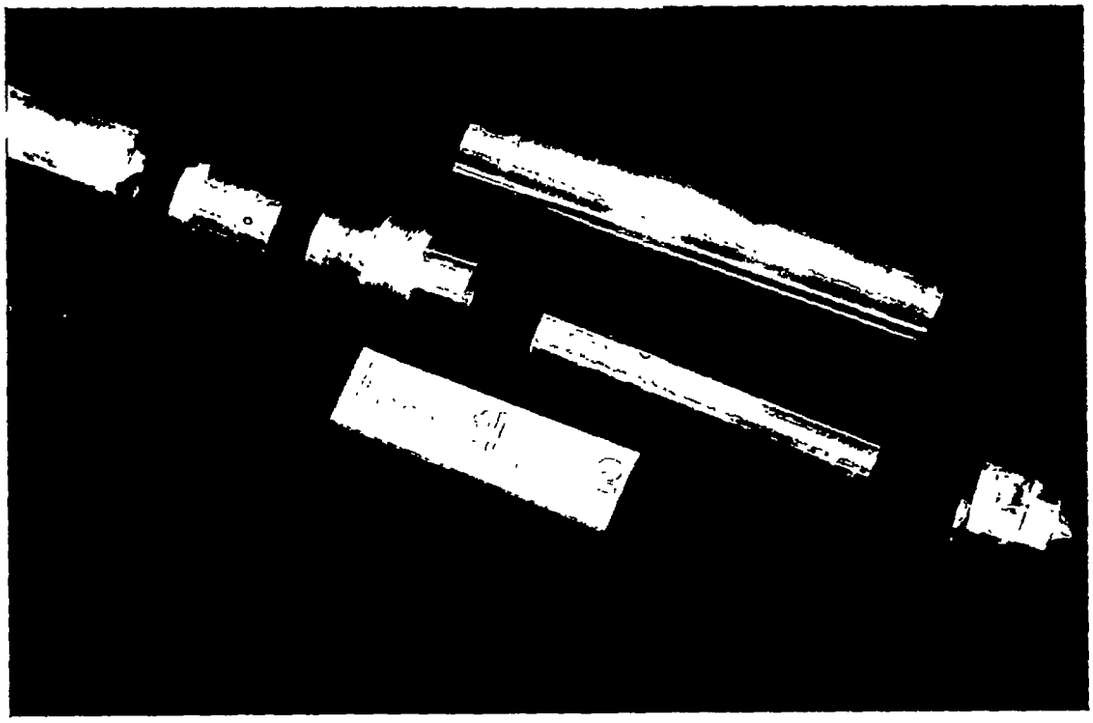
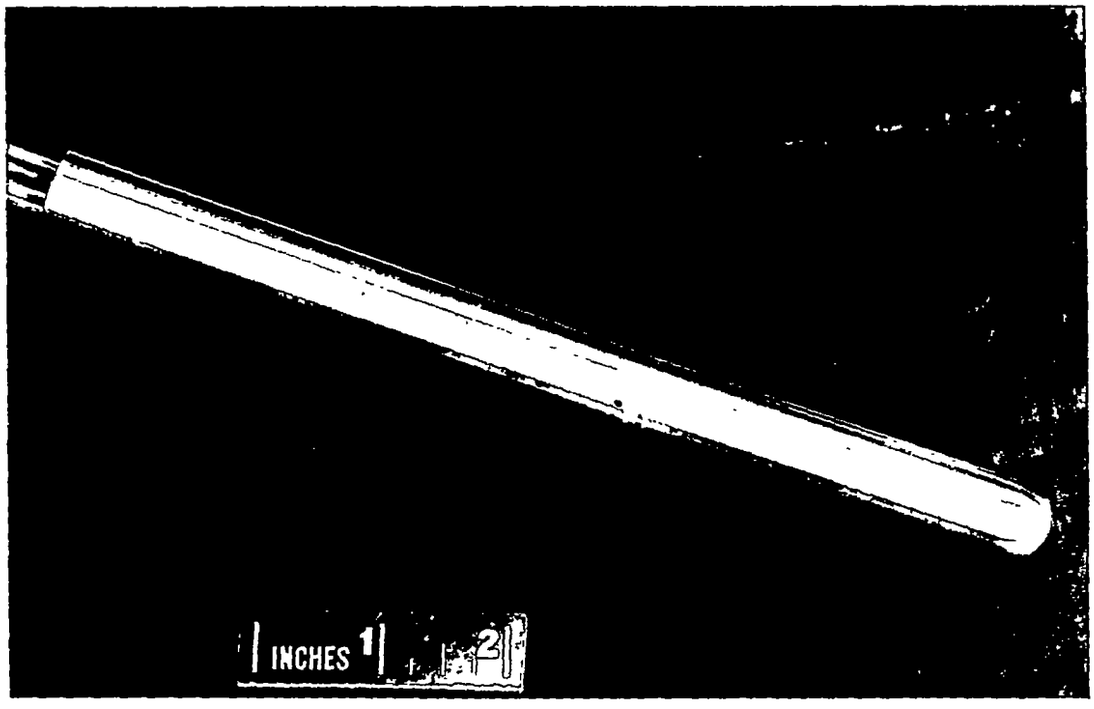


Figure 2.- An exploded view of the sonic-flow temperature chamber and total- and static-pressure probe.



(a) Unassembled. A-21139



A-19800

(b) Assembled.

Figure 3.- Photograph of survey probe.

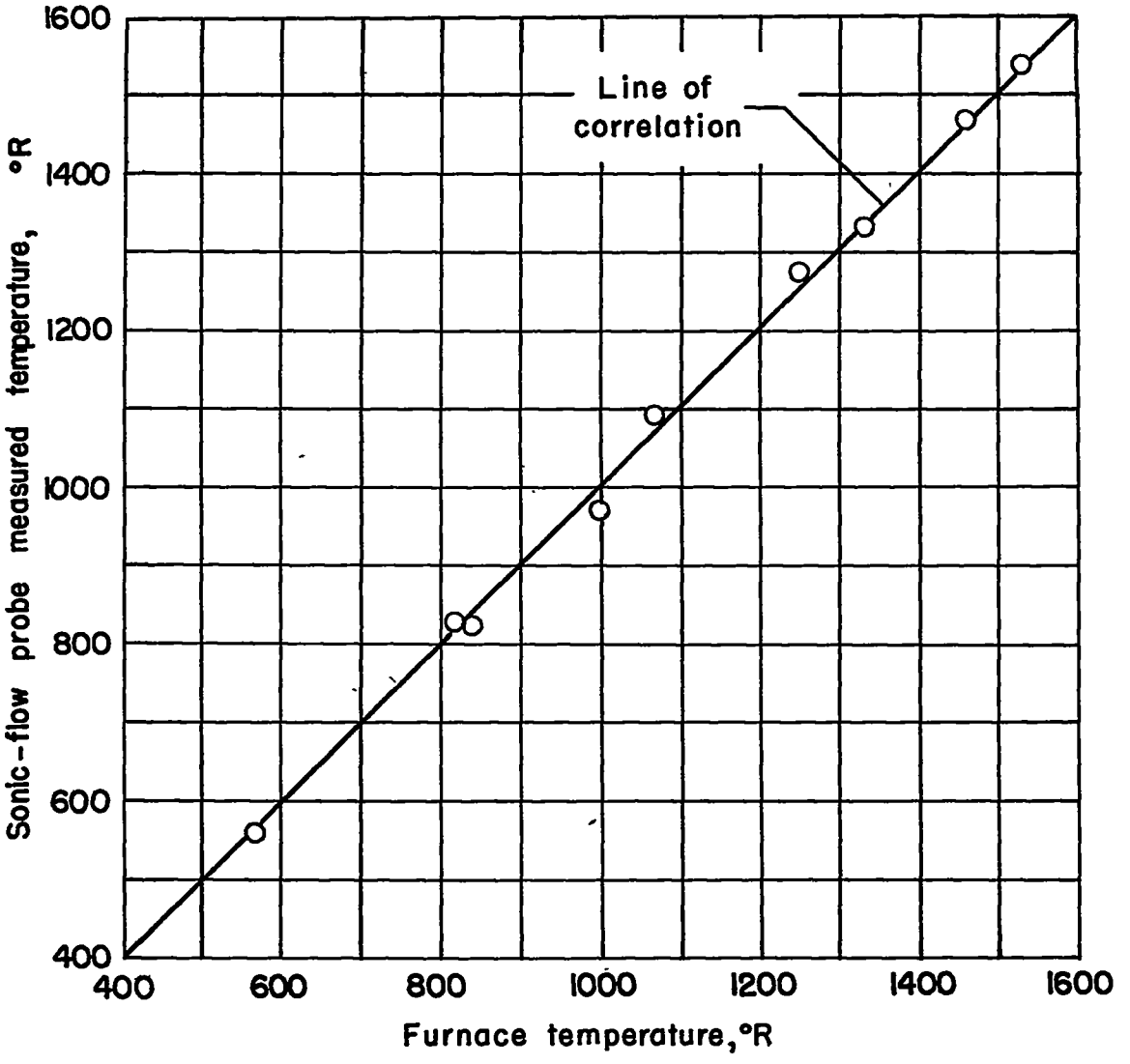


Figure 4.- Static calibration of the sonic-flow probe in a variable temperature furnace.

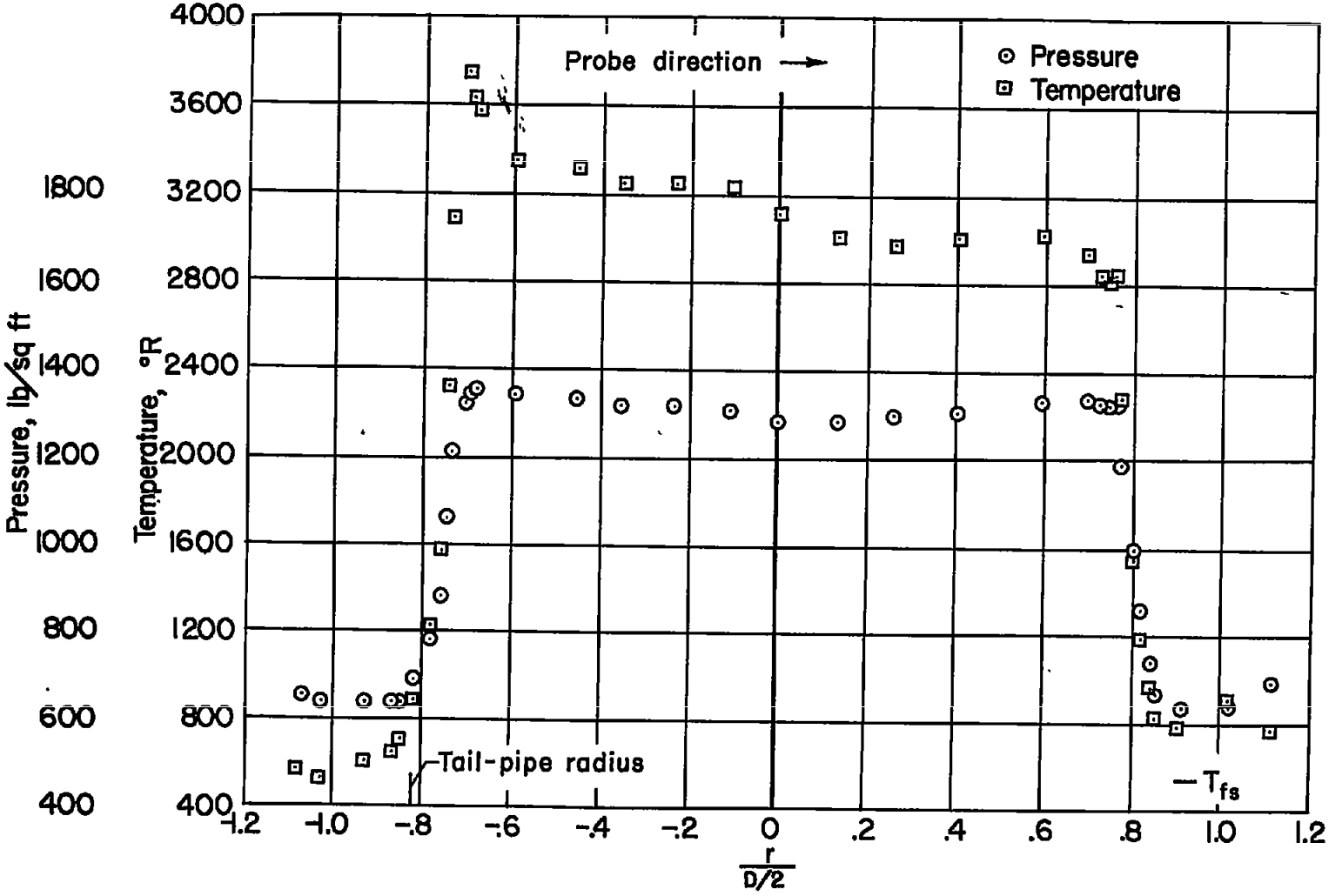


Figure 5.- Total pressure and stagnation temperature profiles; afterburner on, M = 0.63, h_p = 25,000 feet.

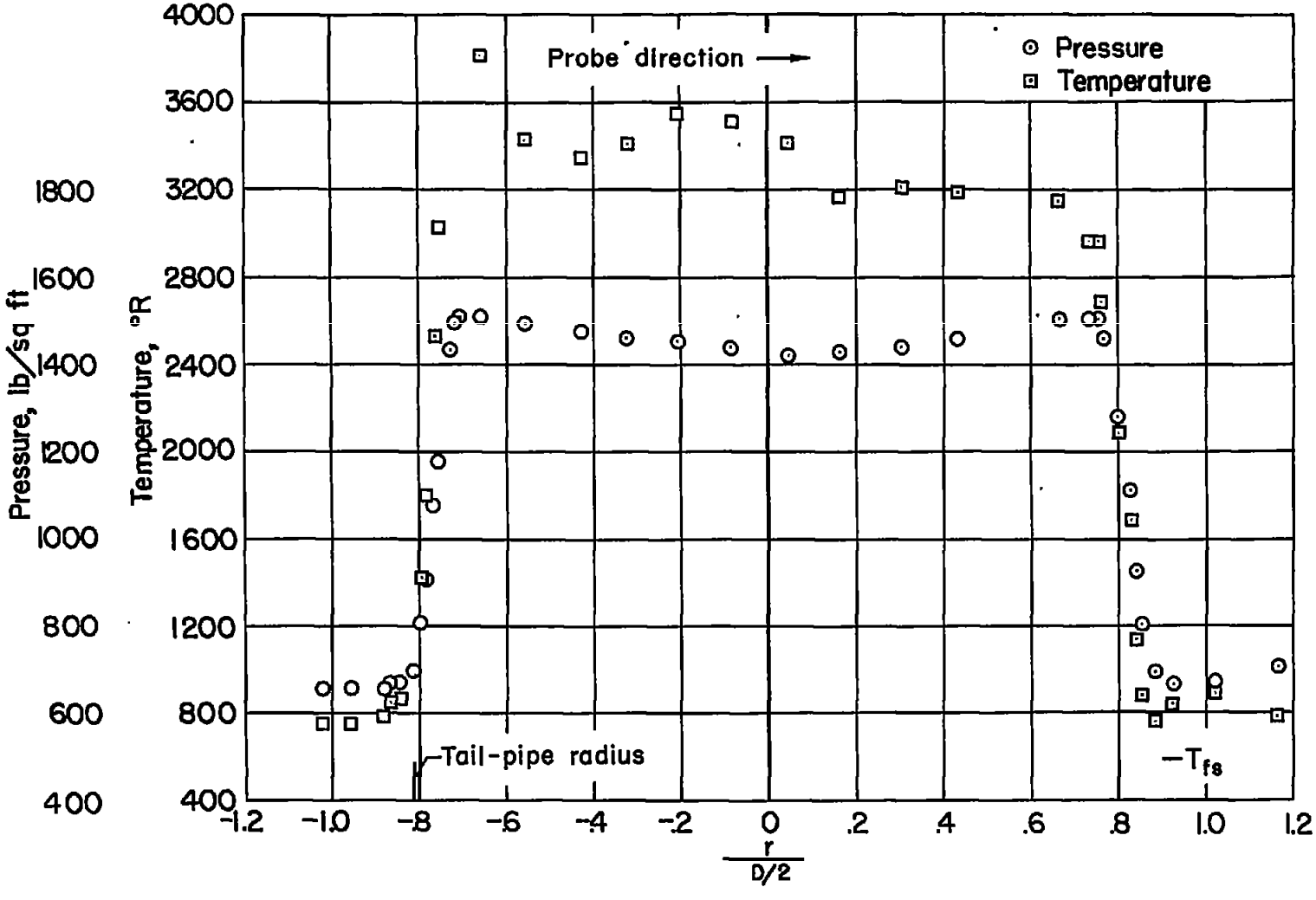


Figure 6.- Total pressure and stagnation temperature profiles; afterburner on, M = 0.82, h_p = 25,000 feet.

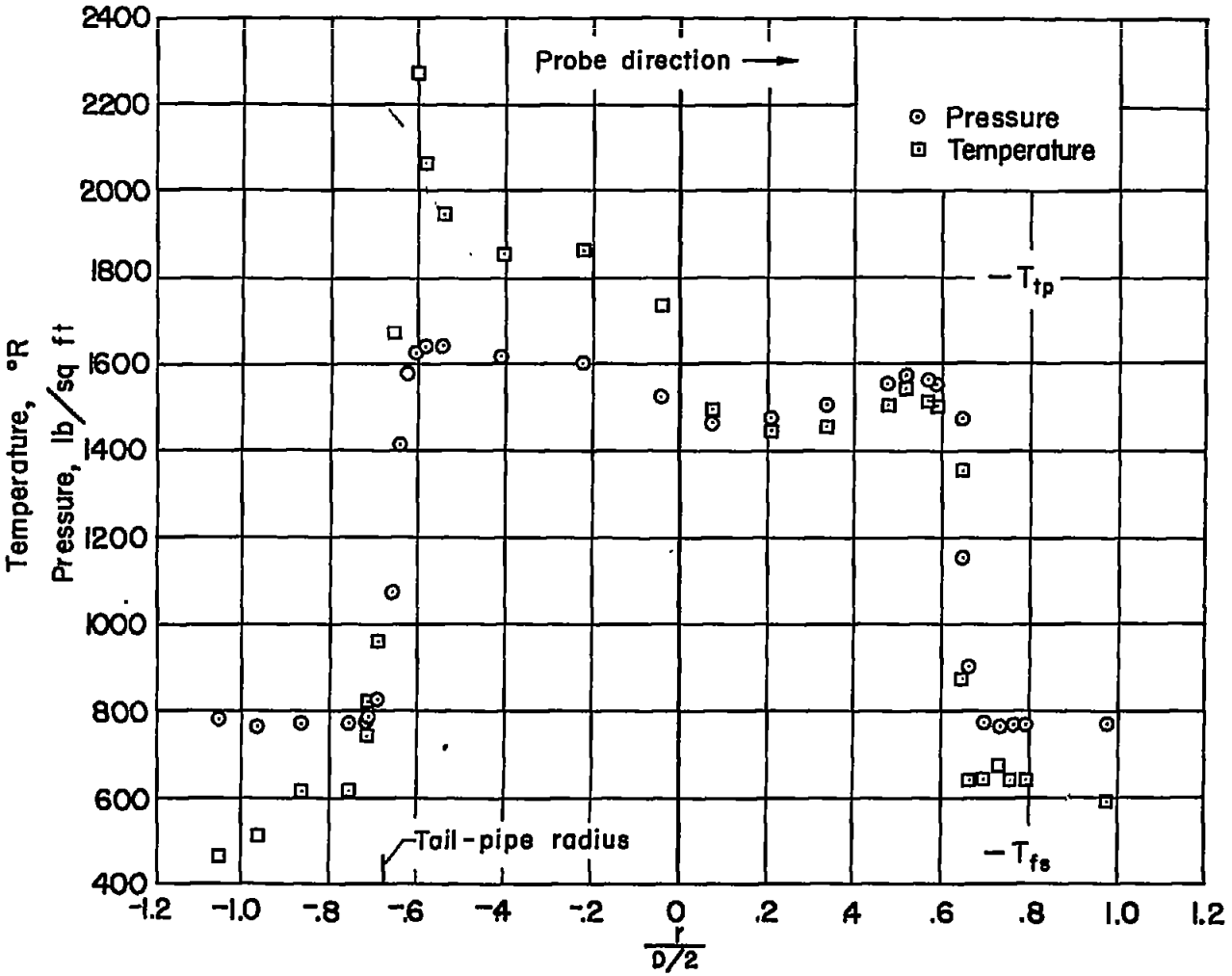


Figure 7.- Total pressure and stagnation temperature profiles; afterburner off, M = 0.63, h_p = 25,000 feet.

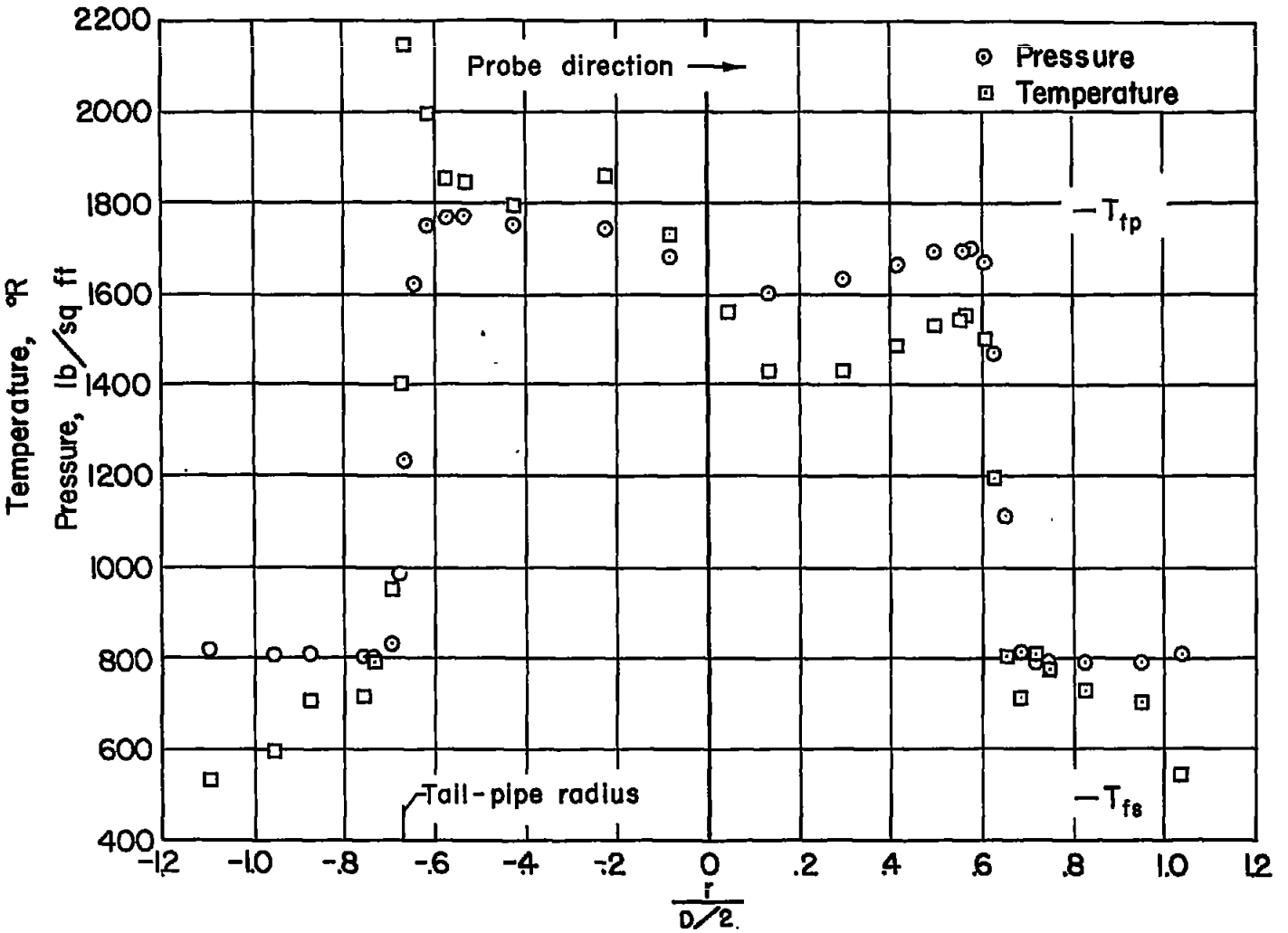
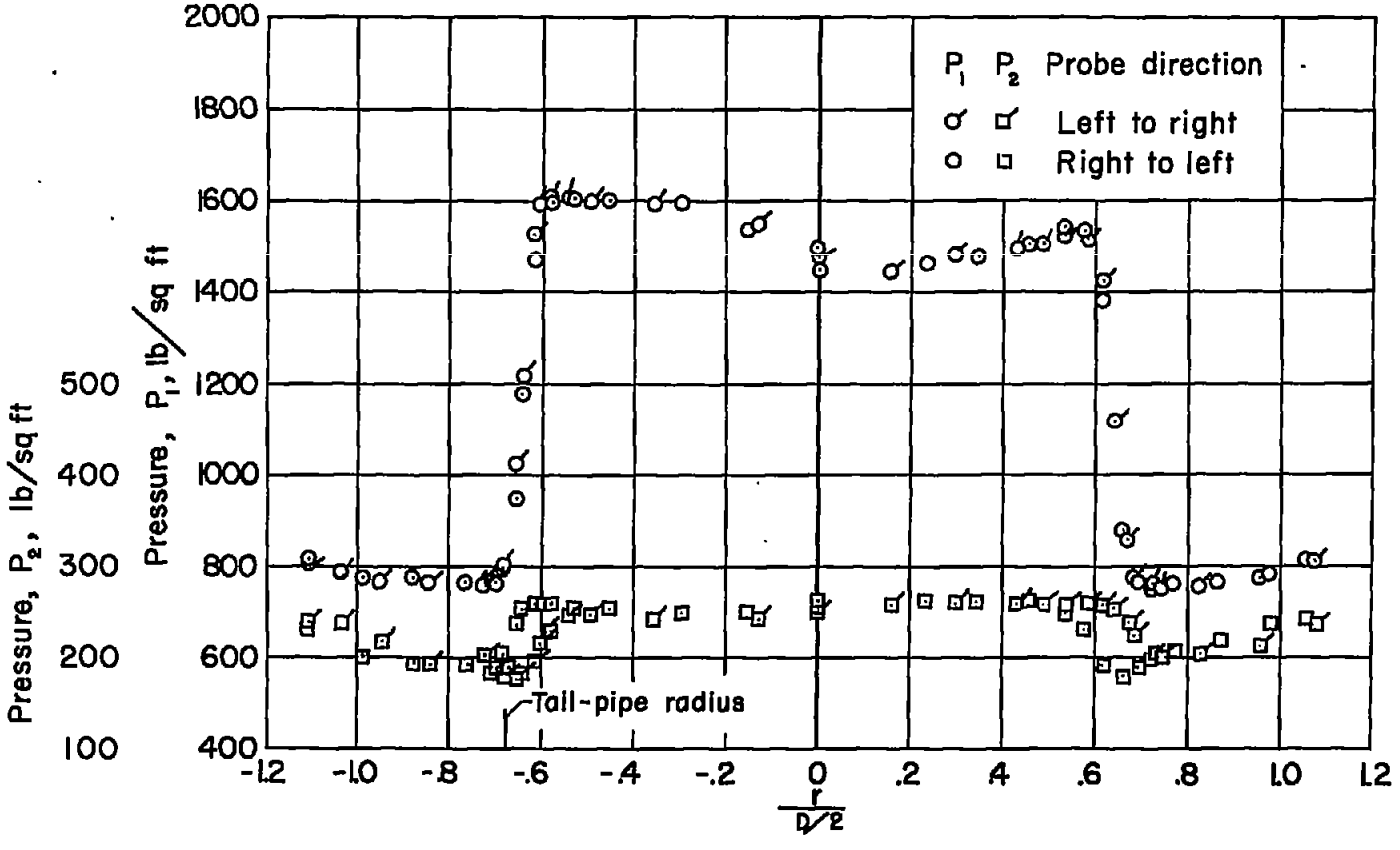
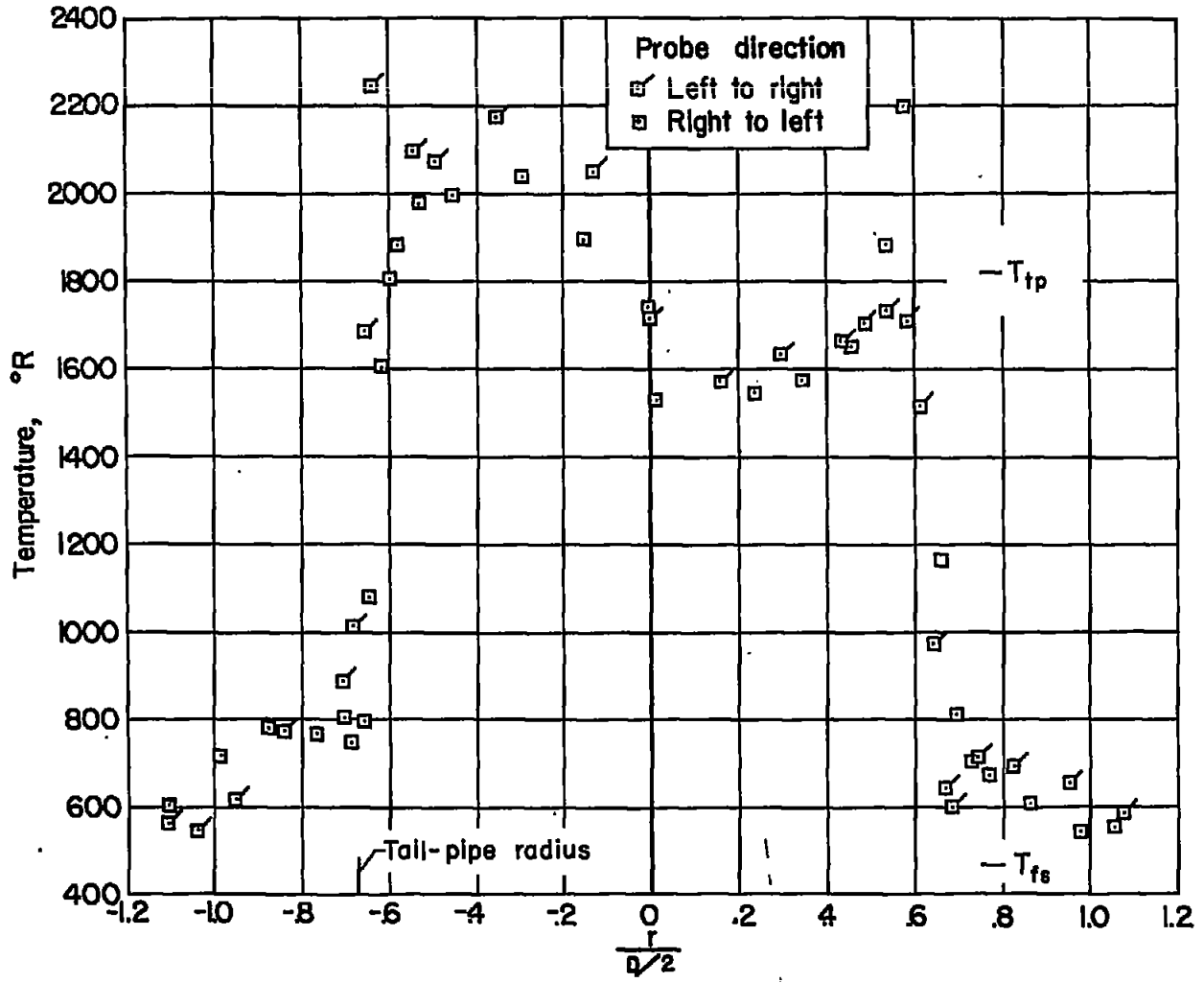


Figure 8.- Total pressure and stagnation temperature profiles; afterburner off, M = 0.80, h_p = 25,000 feet.



(a) Pressure profile.

Figure 9.- Total pressure and stagnation temperature profiles; afterburner off, probe swung both ways, M = 0.55, h_p = 25,000 feet.



(b) Temperature profile.

Figure 9.- Concluded.